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RESEARCH MEMORANDUM

EFFECTS OF CERTAIN FLOW NONUNIFORMITIES ON LIFT, DRAG, AND PITCHING MOMENT FOR A TRANSONIC-AIRPLANE MODEL INVESTIGATED AT A MACH NUMBER OF 1.2 IN A NOZZLE OF CIRCULAR CROSS SECTION

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RESEARCH MEMORANDUM

EFFECTS OF CERTAIN FLOW NONUNIFORMITIES ON LIFT, DRAG, AND PITCHING MOMENT FOR A TRANSONIC-AIRPLANE MODEL INVESTIGATED AT A MACH NUMBER OF 1.2 IN A NOZZLE OF CIRCULAR CROSS SECTION

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SUMMARY

An investigation was conducted in the Mach number 1.2 test section of the Langley 8-foot high-speed tunnel to obtain some quantitative indication of the magnitude of model-force changes produced by a given nonuniformity of the test-section flow. The results of these tests indicate that no significant changes of lift, drag, and pitching-moment coefficients for a test model of a transonic airplane were produced by varying the location of a moderately strong flow disturbance axially along the model length. Also, the approach of the stream normal shock from its usual location downstream of the sting-supported test model is shown to result in no appreciable model-force changes until the fluctuating normal shock reaches the model base and tail surfaces.

INTRODUCTION

The attainment of uniform flow in the test section of a supersonic wind tunnel is very difficult in practice and some nonuniformity is usually present. For some test purposes it is necessary to have the flow as uniform as possible, but for practical force measurements on conventional test models it was thought that some nonuniformity could be tolerated. The present investigation was undertaken to obtain the effects of certain flow disturbances on the force characteristics of a model. For this investigation, a model of a transonic airplane, equipped with an internal balance unit for model-force measurements, was tested in the Mach number 1.2 test section of the Langley 8-foot high-speed tunnel in the presence of a moderately strong compression disturbance. The scope of the measurements was limited to the given flow conditions and to the particular test model.

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Another flow nonuniformity of interest in testing models in a relatively short supersonic test region where the test-section flow is terminated by the stream normal shock is introduced by the proximity of the stream normal shock to the rear of the test model. The shock terminating the Mach number 1.2 test section of the Langley 8-foot high-speed tunnel fluctuated over a considerable axial distance and, therefore, tended to spread its effects over a greater distance than would result from a steady normal shock. Some uncertainty existed as to the distance at which the shock effects might be felt upstream of the average shock location. This uncertainty led to the simple expedient of measuring the forces on a conventional test model while permitting the stream normal shock to approach the base of the model from downstream.

SYMBOLS

L	lift, pounds
D	drag, pounds
M	pitching moment about center of gravity (20.2 percent $\bar{c}$), foot-pounds
$\bar{c}$	wing mean aerodynamic chord, feet
S	surface area of wing, square feet
q	dynamic pressure corresponding to Mach number 1.2, pounds per square foot
C_L	lift coefficient (L/qS)
C_D	drag coefficient (D/qS)
C_m	pitching-moment coefficient $(M/q\bar{c}S)$
α	angle of attack of fuselage center line, degrees

EXPERIMENTAL SETUP

The Langley 8-foot high-speed tunnel in which the present investigation was conducted is of circular cross section throughout. An axially symmetrical nozzle installed as a liner in the tunnel was used for expanding the flow to a Mach number of 1.2, and an effectively cylindrical section about 94 inches in diameter constituted the supersonic test region of parallel flow available for the model tests. The length of this Mach number 1.2 test region was fairly short, varying between 40 and 55 inches; the exact length depended upon various factors affecting the tunnel power. Deceleration of the supersonic flow occurred through a fluctuating normal shock in the stream at the downstream end of the test section. A narrow window extending the length of the nozzle and test section was available for observation of the model and flow phenomena.

A $\frac{1}{16}$-scale model of the Douglas D-558-II airplane complete with sweptback wing and tail surfaces was used for this investigation. This model was equipped with an internal balance unit for measuring model forces. The over-all length of the model fuselage was 31.5 inches and the tail surfaces extended 2.5 inches downstream of the base of the fuselage. The maximum diameter of the fuselage was approximately 3.75 inches and the wing span was 19 inches. The mean aerodynamic chord of the model wing was 5.46 inches and that of the model tail, 2.61 inches. An angle of incidence of 3° and a horizontal-tail setting of 1.9°, both taken relative to the longitudinal axis of the fuselage, and an elevator setting of 0° relative to the horizontal-tail setting were kept constant throughout the tests. The fuselage angle-of-attack variations included only the approximate values of 4.5°, 1.5°, and -2°. The center of gravity of the model was displaced about 2 inches from the nozzle center line for the fuselage angle of attack of 4.5°. A photograph of the test model is shown as figure 1 and a complete description of the model and test equipment is given in reference 1.

The arrangement for testing the model in the supersonic test region of the Langley 8-foot high-speed tunnel is shown in figure 2. The model was supported at the axial center line of the tunnel by means of a conical sting extending from the base of the model to a cylindrical tube which was maintained in place along the center line by a support system located in the tunnel diffuser. Adjustment of the axial location of the model was accomplished by axial movement of the cylindrical tube inside the central hub of the support system. Model angle-of-attack changes were effected by means of interchangeable split couplings at the downstream end of the conical sting. Corrections for the model angle-of-attack changes due to bending of the sting and extension tube during actual tests were obtained from the angle of reflection of a beam of light projected obliquely from the test-section observation window to a small mirror imbedded in the surface of the test model. Static-pressure orifices located 2 inches apart along the walls of the nozzle and test section were used to check the flow Mach numbers obtained for the various runs.

Surveys of the axially symmetrical flow in the supersonic test section at the time of the present investigation indicated that the flow was relatively free of serious disturbances, except at a

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station on the axial center line about 75 inches downstream of the effective minimum section of the nozzle. At this station, a compression disturbance equivalent to a decrement of 0.05 in the Mach number 1.2 flow persisted over an axial distance of about 2 inches. This disturbance was found to extend obliquely out from its central location along lines roughly corresponding to Mach lines. The intensity of the disturbance decreased at a significant rate with distance from the axial center line, the Mach number decrement of 0.05 from the center line to 1.5 inches off the center line decaying to decrements of 0.038 and 0.022 at distances of 3 and 7 inches off the center line, respectively. This rate of decay of the flow-disturbance intensity with distance from the center line indicated that the magnitude of the disturbance actually striking the surface of the model fuselage might be somewhat less than the values shown in figure 3, since these measurements were made at a distance of 1 inch off the axial center line and the surface of the fuselage was as much as 1.9 inches from the center line. Axial distributions of the flow Mach number 1 inch off the center line of the Mach number 1.2 test section are given in figure 3 for model locations designed to locate the central disturbance roughly in regions occupied by the model wing, by the model tail, and by the fuselage between the wing and tail.

TEST PROCEDURE

The calibration Mach number distribution in the supersonic test section of the Langley 8-foot high-speed tunnel was held constant for the tests of this investigation. The given compression disturbance equivalent to a decrement of 0.05 in the flow at the center line persisted at a fixed axial location in the test region of Mach number 1.2. The axial location of the test model was varied to permit the flow disturbance to strike the model at various longitudinal stations and thus to vary the uniformity of the flow over the model components, especially in the regions of the wing and tail surfaces. Forces were measured for the model in the three regions of varying flow uniformity shown in figure 3.

Reynolds numbers for the model fuselage, wing, and tail, based on test conditions at Mach number 1.2 and on the over-all fuselage length and mean aerodynamic chords of the wing and tail, were approximately 10×10^6, 1.74×10^6, and 0.83×10^6, respectively.

The stream normal shock at the downstream end of the Mach number 1.2 test section fluctuated rapidly over a distance of about 8 inches in the stream direction. This shock, normally located about 12 inches downstream of the model, was permitted to approach the base of the test model from downstream by suitable reduction of the tunnel operating power. Model forces were measured with the shock at various distances downstream of the base of the model. The average shock locations were determined by means of shadow observations and by means of static-pressure measurements at the test-section wall. No static-pressure measurements were made on the sting and no attempts were made to separate possible effects caused by the normal shock striking the model from those due to transmission of increased pressures upstream through the boundary layer on the sting.

RESULTS AND DISCUSSION

Varying flow uniformity longitudinally along length of model.- Model-force measurements with the test model in the flow distributions of figure 3 are presented in figure 4. These results indicate that no significant changes of the lift, drag, and pitching-moment coefficients were produced by varying the axial location of the test model in the Mach number 1.2 flow containing a compression disturbance which extended over an axial distance of 2 or 3 inches and which was equivalent to a decrement of 0.05 in the flow Mach number at the axis of symmetry. The variation of the force coefficients shown in figure 4 is within the accuracy of measurement normally attainable by means of the internal balance system used for measuring model forces. Significant changes of model forces due to variation of the flow uniformity may possibly occur, if the model is tested in regions where the force coefficients vary rapidly with Mach number.

The experimental indication that no significant changes of model lift, drag, and pitching moment are produced by the given flow disturbance does not necessarily imply that such a disturbance can be tolerated for more fundamental studies of aerodynamic phenomena.

Nearness of stream normal shock to base of model.- The results of force measurements for the test model with the stream normal shock located various distances downstream of the model base are shown in figure 5. The normal-shock locations in this figure are given in terms of the distance from the model base to the mean shock position. The shock actually fluctuates about 4 inches upstream and downstream of this mean position. The results shown in figure 5 indicate that significant changes of model drag and pitching-moment coefficients are introduced when the average location of the stream normal shock is at the base of the model and that these changes diminish in magnitude as the fluctuating shock is moved downstream, until no appreciable changes are evident when the shock-fluctuation zone is entirely downstream of the model base and tail surfaces. The model lift coefficient is shown to remain essentially constant throughout these tests (see fig. 5) and, therefore, indicates that the flow in the region of the wing was unaffected by the fluctuating normal shock even when the shock was located at the base of the model.

Whether the model-force changes were due solely to primary effects of the stream normal shock in the region of the test model or whether some portion of the over-all changes was possibly brought about through transmission of the increased pressures behind the shock upstream through the boundary layer on the sting is not known. It is believed, however, that no large changes in model forces are introduced by the transmission of increased pressures upstream through the sting boundary layer when the shock is located entirely downstream of the model. This belief is supported by the fact that the measured drag and pitching-moment coefficients tend to level off at constant values when the fluctuating-shock zone is entirely downstream of the model. (See fig. 5.)

CONCLUSIONS

The effects of certain flow disturbances on model forces were investigated for a given set of conditions. The results of the brief investigation appeared to justify the following conclusions regarding effects of given flow nonuniformities on force measurements for a test model of a conventional transonic airplane in an axially symmetrical flow of Mach number 1.2:

1. A flow disturbance equivalent to a Mach number decrement of 0.05 or less at the axis of symmetry and extending over an axial distance of 2 or 3 inches or 6 to 10 percent of the model length can be tolerated for practical force-test purposes.

2. No significant changes of model drag and pitching-moment coefficients occurred until the stream normal shock was located in the region of the model base and tail surfaces.

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REFERENCE

1. Osborne, Robert S.: High-Speed Wind-Tunnel Investigation of the Longitudinal Stability and Control Characteristics of a $\frac{1}{16}$-Scale Model of the D-558-2 Research Airplane at High Subsonic Mach Numbers and at a Mach Number of 1.2. NACA RM L9C04, 1949.

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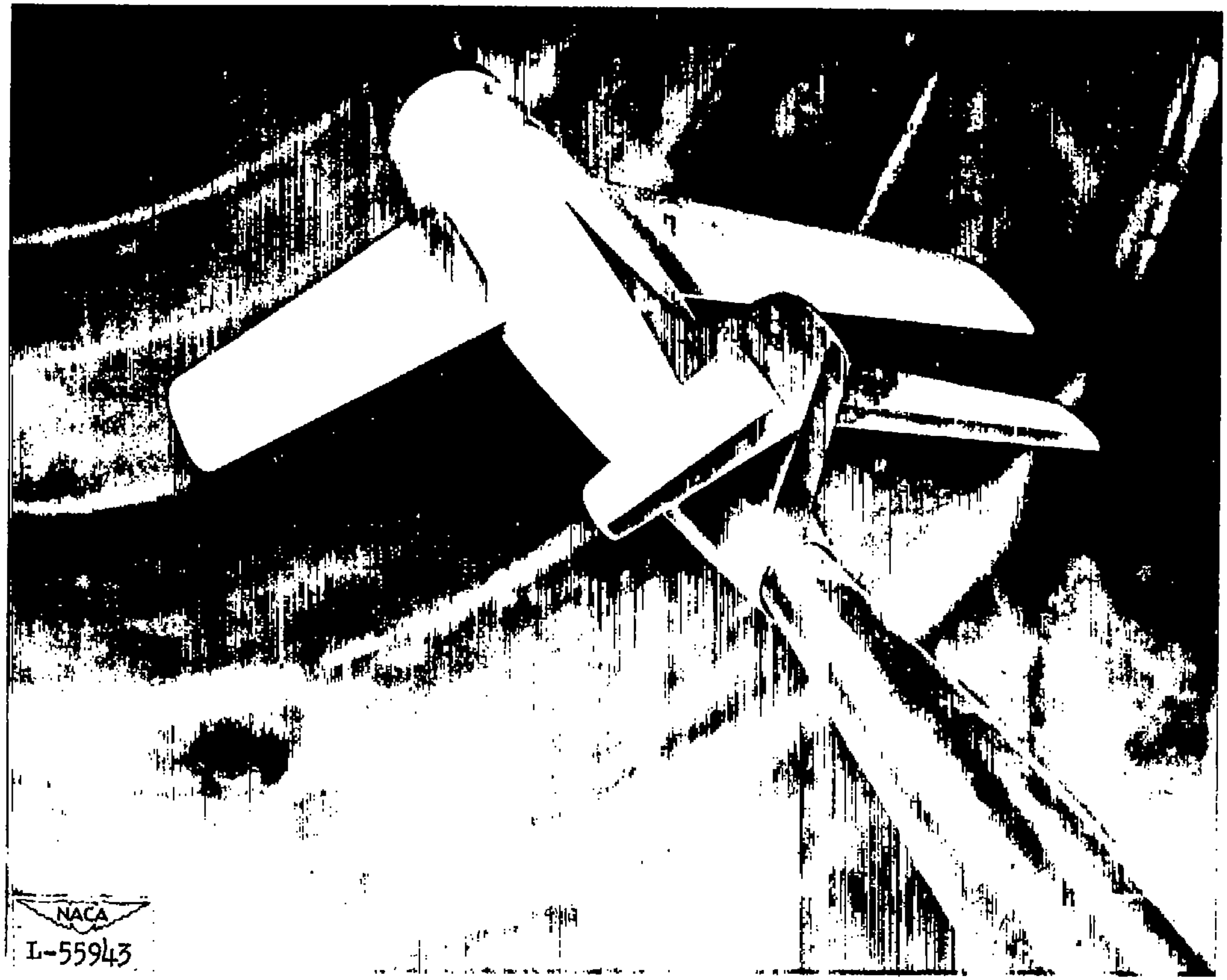

Figure 1.- Sting-supported $\frac{1}{16}$-scale model of the D-558-II airplane used for investigating the effects of flow nonuniformity on model forces.

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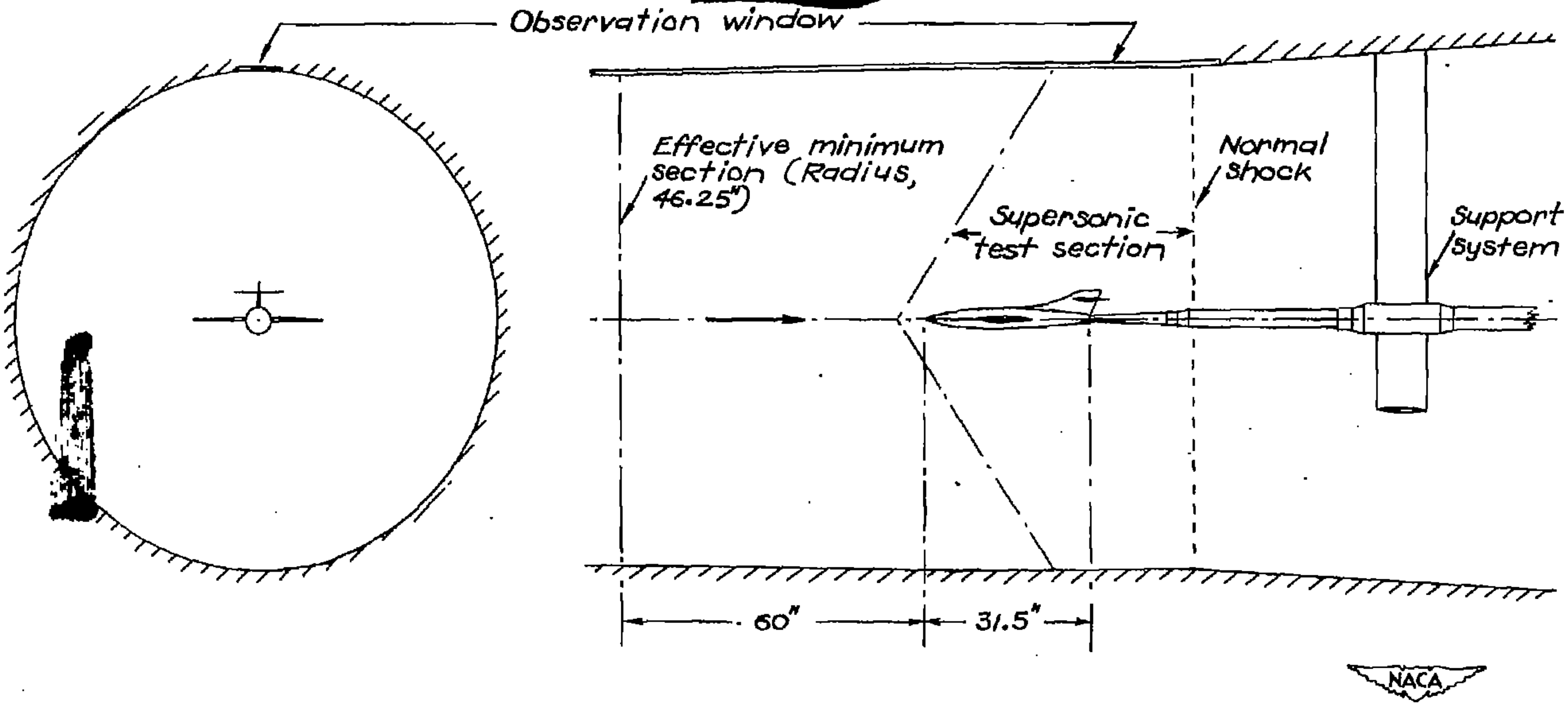

Figure 2.- Arrangement for testing a complete airplane model at Mach number 1.2 in the Langley 8-foot high-speed tunnel.

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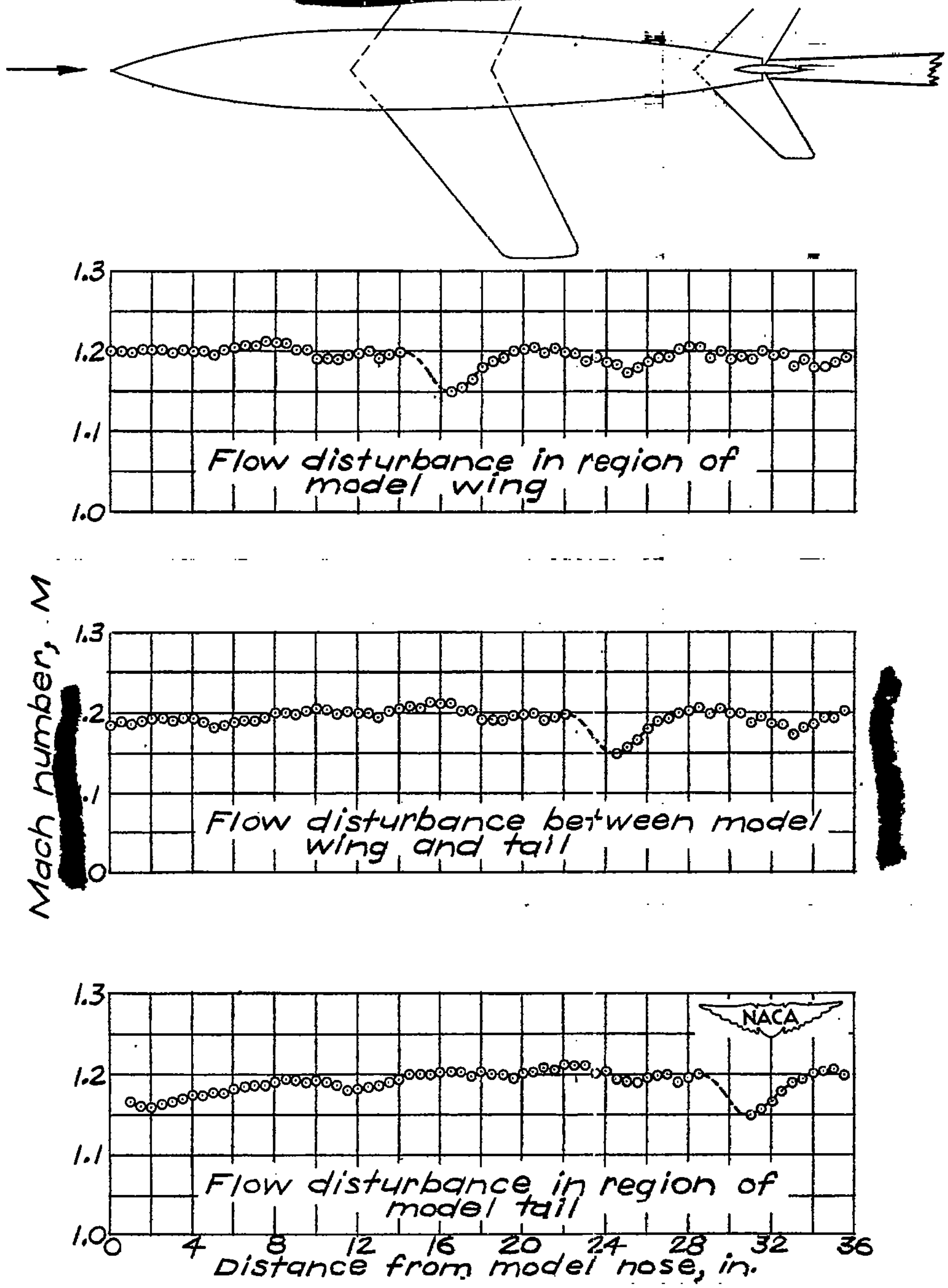

Figure 3.–Free-tunnel Mach number distributions 1-inch from tunnel center line at three locations subsequently occupied by the model.

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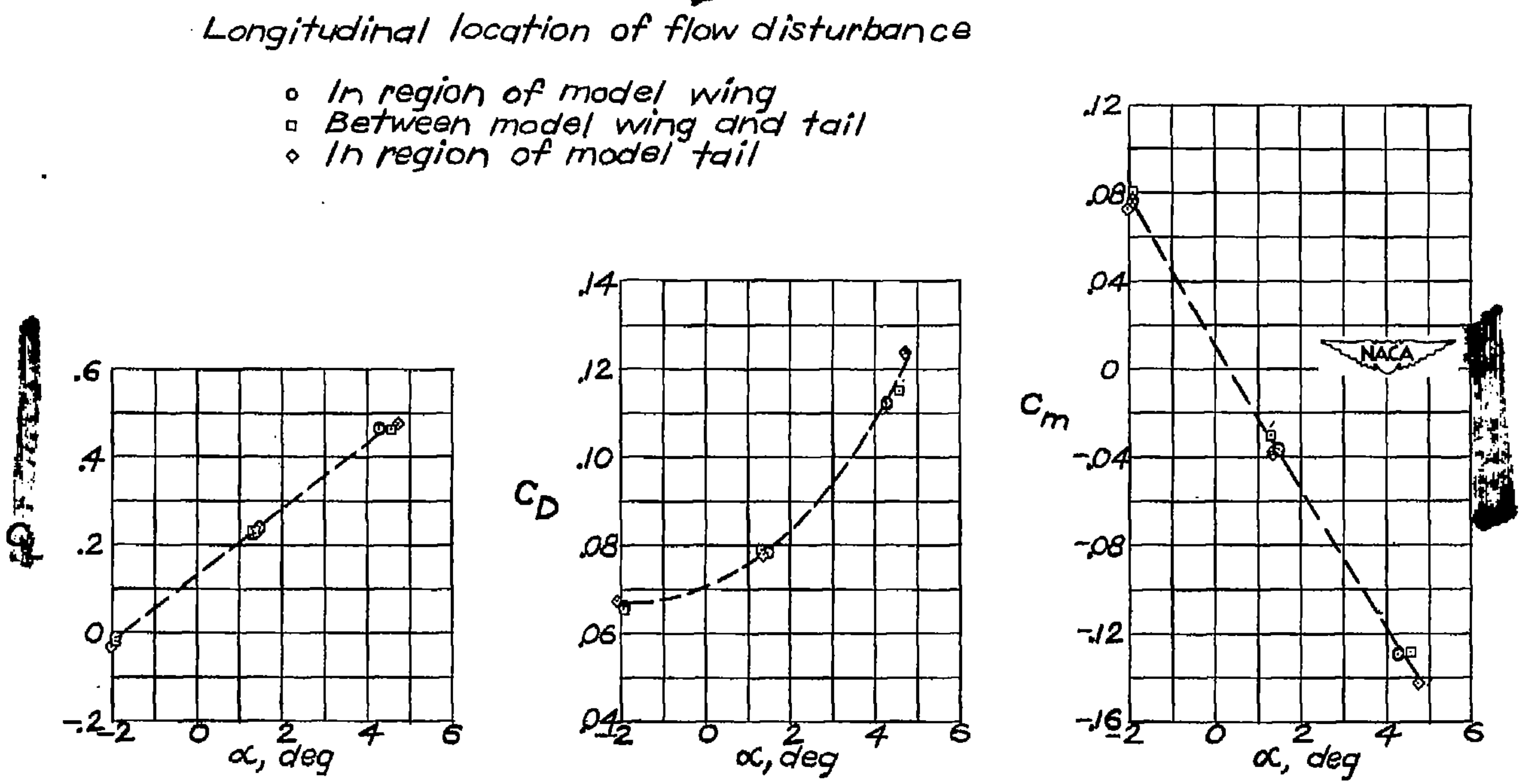

Figure 4.– Lift, drag, and pitching-moment coefficients of the model for three positions of the flow disturbance along the length of the model.

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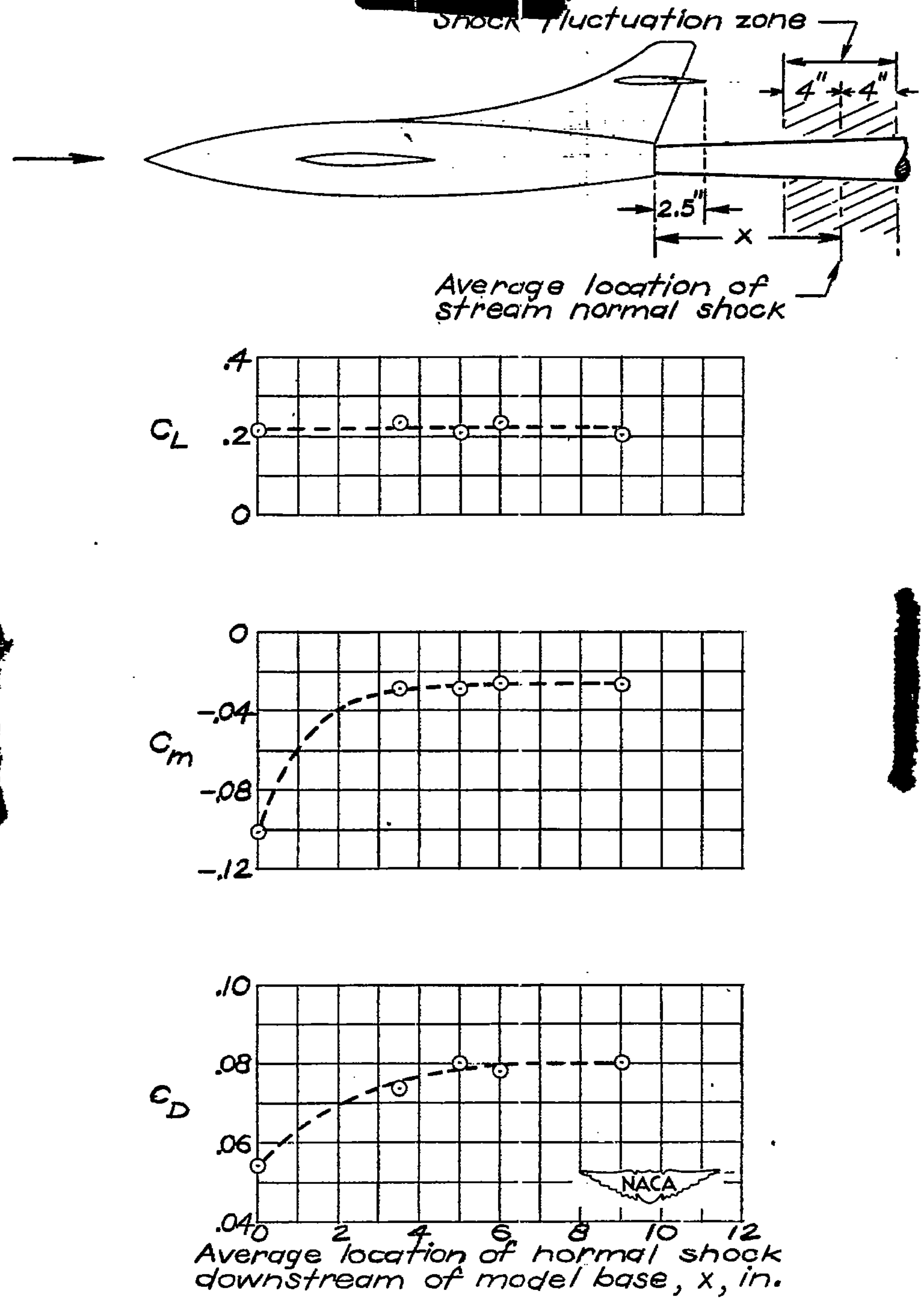

Figure 5.- Lift, drag, and pitching-moment coefficients for the model at various locations of the stream normal shock downstream of the base of the model.

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